ROTATING COMBUSTOR GAS TURBINE ENGINE

FIELD OF THE INVENTION

[001] The present invention relates to gas turbine engines. Even more particularly, this invention relates to gas turbine engines having improved efficiency, high-power density and low manufacturing costs.

BACKGROUND OF THE INVENTION

[002] The gas turbine engine has become one of the primary devices used to convert chemical energy in the form of fuel into mechanical energy. Since its invention the performance of the gas turbine engine has continued to improve, delivering high efficiency, high power density and low emissions of environmental pollutants. Gas turbine engines are extensively used to power large aircraft, for large marine propulsion applications and also by utilities to generate electricity. Due to certain design limitations, however, gas turbine engines have not been extensively used to power automobiles, or for small marine or private aircraft applications, nor have gas turbines been used extensively for distributed power generation systems.

[003] A traditional prior art gas turbines engine consists of four primary components; a compressor, a diffuser, a combustor and an expander. The compressor takes in ambient air, compresses it to a higher total pressure and temperature, and then discharges it at a high speed to the diffuser. The high-speed flow from the compressor is slowed in the diffuser to a lower velocity, which is required for combustion. In the combustor fuel is injected, mixed and burned creating a hot gas, which then expands across the turbine creating a certain quantity of power. The turbine supplies the power required to drive the compressor. Excess power can then be used to drive an external load such as producing thrust in an aircraft application or running a generator in an electrical power application.

[004] Fluid dynamic scaling and cost considerations have traditionally favored large size engines. Specifically, large gas turbine engines, such as those used for power generation, rely

on high combustor exit and turbine inlet temperatures and precise dimensional control to achieve high combustion and component efficiency. Maintaining high temperatures is important because its sets the amount of energy that is deposited in the flow, and therefore limits the maximum power than can be extracted from the device. Higher temperatures also lead to higher overall cycle efficiencies.

[005] The power densities and efficiencies required of today's most demanding applications (aircraft propulsion and stationary power generation) are such that the flow temperatures significantly exceed the melting temperatures of the materials comprising the engine. Therefore, sophisticated cooling systems and fabrication techniques are required to accommodate these high temperatures. The cost and difficulty in achieving high component machine tolerances and accommodating high temperatures in relatively smaller sized conventional engines have made gas turbines a less attractive power source for power levels less than hundreds of kilowatts.

[006] As an example, high performance gas turbines use high efficiency axial flow compressors and cooled axial flow turbines. Axial flow compressors require many stages and precise dimensional control to achieve high efficiencies and high-pressure ratios. This precise dimensional control is expensive to reproduce for smaller sized engines. In addition, axial flow turbines are cooled by extracting a portion of the compressor discharge air upstream of the combustor and using this air to cool the turbine by routing the air through cooling passages inside the turbine components. The cooling air then discharges through film-cooling holes in the surfaces of the turbine components where it forms a protective cool film over the surfaces of the flow-path there by reducing the component temperature. These flow passages are also difficult and expensive to reproduce in smaller sized engines.

[007] To reduce manufacturing costs, small gas turbine engines generally use radial flow compressors and either radial inflow or axial flow uncooled turbines. While radial flow compressors are generally less efficient than axial compressors, this does not represent a fundamental limitation for high efficiency small gas turbines. The real drawback with small gas turbine engines is that the maximum allowable material temperature of the uncooled turbine limits the turbine entry temperature. Low turbine entry temperatures result in low engine efficiencies. In certain configurations, gas turbines can use exhaust gas regenerators to compensate for the relatively low turbine entry temperatures; however, this adds

considerable manufacturing expense to the gas turbine and significantly reduces the engine power density.

[008] What is needed is a high efficiency gas turbine engine that is simple in design and that can be economically produced on a smaller scale.

[009] Other objects will, in part, be obvious and will, in part, appear hereinafter. The invention accordingly, comprises the features of construction, combination of elements and arrangements of parts, which will be exemplified in the following detailed description and the scope of the invention will be indicated in the claims.

SUMMARY OF THE INVENTION

[0010] The present invention achieves these and other objectives by providing a rotating combustor gas turbine engine system that combines the afore-mentioned separate functions of conventional prior art gas turbine engines into a single rotating device.

[0011] According to one aspect of the invention, a gas turbine engine comprises a rotating combustion system and a rotating turbine system operatively connected to one another to form a single rotating assembly that rotates about a common axis.

[0012] As to another aspect of the invention, the gas turbine engine includes a rotating nozzle and a downstream stationary diffuser or a counter rotating stage.

[0013] As to another aspect of the invention, the gas turbine engine includes an upstream compressor.

[0014] As to a further aspect of the invention, the gas turbine engine includes a combustion manifold, a turbine nozzle and a series of turbine blades. The turbine blades may extend from the combustion manifold to the turbine nozzle.

[0015] As to yet another aspect of the invention, the combustor and turbine of the gas turbine engine operate in a radially outflow configuration. In this and other configurations, the gas turbine engine may also include an axial outflow nozzle.

[0016] According to yet another aspect of the invention, the compressor includes an inducer to provide for a pressure rise.

[0017] As to another aspect of the invention, the gas turbine engine includes a secondary impeller for isolating a premised fuel mixture from a series of heated upstream walls.

[0018] As to another aspect of the invention, a fuel is directly injected into a combustion chamber in such a manner as to produce a temperature gradient that isolates a series of turbine blades from a hot combustion gas.

[0019] As to another aspect of the invention, the gas turbine engine includes a cooling system designed to cool the components of the gas turbine engine. The cooling system may include a closed loop through which a liquid flows and or a cooling fluid that undergoes a liquid-vapor phase change, thereby cooling the walls of the combustor.

[0020] As to yet a further aspect of the invention, the gas turbine engine includes an ignition system for igniting a fuel/air mixture. The ignition system may include a resistive heater, a spark ignition, a pilot flame and or a pre-heater.

[0021] As to a further aspect of the invention, the single rotating component may operate as a core to a multi-stage gas turbine. In this and other embodiments the gas turbine engine may include at least one additional axial or radial compressor stage upstream of the single rotating component and or at least one additional axial or radial turbine stage downstream of the single rotating component.

[0022] As to another aspect of the invention, the gas turbine engine includes a counter-rotating turbine operatively connected to turbine.

[0023] As to another aspect of the invention, the gas turbine engine includes a rotating nozzle. The rotating nozzle may utilize a convergent-divergent geometry.

[0024] Another aspect of the invention is directed to a gas turbine engine that includes a compressor, a turbine operatively connected with the compressor into a single impeller having a plurality of rotating passages and a combustion system integrated into the rotating passages of the compressor/turbine single impeller.

[0025] As to another aspect of the invention, a gas turbine engine includes a compressor, a combustor and a turbine, wherein the compressor, combustor and turbine are operatively integrated into a single rotating assembly. In certain embodiments, the compressor, combustor and or turbine are separately removable.

[0026] Another aspect of the invention is directed to a gas turbine engine comprising a rotating combustion system, a radial impeller operatively connected to the rotating combustion system and a rotating turbine nozzle having a series of turbine nozzle blades, wherein the rotating combustion system remains geometrically fixed in relation to the radial impeller and turbine nozzle blades, all of which spin at a similar rate of rotation about a common axis.

[0027] As to another aspect of the invention, the gas turbine engine includes a stationary diffuser or counter-rotating downstream turbine stage. In this and certain other aspects of the invention, a flow may enter the rotating combustion system from the radial impeller, exit the rotating combustion system into the rotating turbine nozzle while in a common rotating reference frame and then exit the common rotating reference frame where it would be diffused in the stationary diffuser or counter-rotating downstream turbine stage

BRIEF DESCRIPTION OF THE DRAWINGS

[0028] The preferred embodiments of the invention will hereinafter be described in conjunction with the appended drawings provided to illustrate and not to limit the invention, where like designations denote like elements, and in which:

[0029] FIG. 1 is a cross-sectional view of a rotating combustor gas turbine system in accordance with one embodiment of the present invention;

[0030] FIG. 2a is an enlarged fragmentary perspective view of a first side of the rotating combustor gas turbine system of FIG. 1;

[0031] FIG. 2b is an enlarged fragmentary perspective view of a second side of the rotating combustor gas turbine system of FIG. 1;

[0032] FIG. 3 is a further enlarged fragmentary perspective view of the rotating combustor gas turbine system of FIG. 2b;

[0033] FIG. 4 is a further enlarged fragmentary perspective view of the rotating combustor gas turbine system of FIG. 2b;

[0034] FIG. 5 is a cross-sectional view of the rotating combustor gas turbine system of FIG. 1;

[0035] FIG. 6 is an enlarged fragmentary cross-sectional view of the rotating combustor gas turbine system of FIG. 5 taken along line A—A;

[0036] FIG. 7 is an enlarged fragmentary perspective view of a rotating combustor gas turbine system in accordance with a second embodiment of the invention;

[0037] FIG. 8 is a cross-sectional view of a rotating combustor gas turbine system in accordance with a third embodiment of the invention;

[0038] FIG. 9 is an enlarged cross-sectional view of the rotating combustor gas turbine system of FIG. 8;

[0039] FIG. 10 is an enlarged cutaway perspective view of the rotating combustor gas turbine system of FIG. 8;

[0040] FIG. 11 is a cross-sectional view of a rotating combustor gas turbine system in accordance with another embodiment of the present invention;

[0041] FIG. 12 is a cross-sectional view of a rotating combustor gas turbine system in accordance with another embodiment of the present invention;

[0042] FIG. 13 is an enlarged fragmentary perspective view of the rotating combustor gas turbine system of 11;

[0043] FIG. 14 is an enlarged cutaway perspective view of the rotating combustor gas turbine system of FIG. 11;

[0044] FIG. 15 is a cross-sectional view of a rotating combustor gas turbine system in accordance with another embodiment of the present invention;

[0045] FIG. 16 is an enlarged fragmentary cross-sectional view of the rotating combustor gas turbine system of FIG. 15 taken along line B—B;

[0046] FIG. 17 is a cross-sectional view of a rotating combustor gas turbine system in accordance with another embodiment of the present invention; and

[0047] FIG. 18 is a fragmentary cross-sectional view of a rotating combustor gas turbine system in accordance with another embodiment of the present invention.

DETAILED DESCRIPTION OF THE PREFERRED EMBODIMENTS

[0048] Preferred embodiments of the present invention are illustrated in FIGS. 1-18. Referring now to FIG. 1, a rotating combustor gas turbine engine, generally designated by the numeral 10, is shown, wherein the afore-mentioned compressor, combustor and turbine functions of the prior art gas turbine engines are combined into a single rotating device.

[0049] In summary, the rotating combustor gas turbine engine 10 includes an impeller inlet port 12 and a rotating compressor system 14 for compressing an incoming airflow 16, with a rotating combustion system 18 located to receive compressed air from the rotating compressor system 14. A rotating turbine system 20 receives compressed high temperature gas 152 from the rotating combustion system 18. A fixed exit diffuser 24 then increases the static pressure of the high velocity flow from the rotating turbine system 20 using a plurality of static radial exit guide vanes 26 [26 is not shown in fig 1]. The flow is then vented out of the gas turbine engine 10 through an exhaust manifold 28.

[0050] FIG. 2a shows in detail a section through the rotating combustor gas turbine 10 as viewed from the upstream or inlet side of the device, while Figure 2b shows a section through the same device as viewed from the downstream or exit side of the device. The rotating combustor gas turbine 10 includes a one-piece compressor/turbine impeller, generally represented by the numeral 30, comprising an impeller shaft 40, a plurality of compressor blades 42, an impeller disk 44 and a plurality of turbine blades 46.

[0051] In the present embodiment the rotating compressor system 14, comprising an impeller shroud 50, the compressor blades 42 and the impeller disk 44, operates to form a series of

compressor flow paths 34. The impeller disk 44, the turbine blades 46 and a turbine shroud 52 operate to form a turbine flow path 60. In operation, the incoming airflow 16 enters the rotating combustor gas turbine 10 through the impeller inlet port 12. The airflow 16 is then redirected radially and compressed as it flows outward through a series of compressor flow passages 34, created between the compressor blades 42.

[0052] In this configuration the rotating compressor system 14 pressurizes the incoming flow. The inlet port 12 of the rotating compressor system 14, also known as an inducer, reduces the high velocity inlet flow 16 to a lower velocity, higher static pressure flow. The flow 16 is then further pressurized as it flows radially outward in the rotating compressor system 14. As the ratio of the inlet-to-tip radius increases a greater proportion of the impeller pressure rise is generated in the inducer. If the inlet radius equals the impeller tip radius all of the pressure rise will come from the inducer. In an alternative embodiment, a compressor configuration that utilizes an inducer to provide the entire pressure rise for the compressor system is another possible compressor configuration.

[0053] Fuel 62 is brought into the impeller 30 through a fuel inlet channel 64 in the impeller shaft 40. The fuel 62 is injected as the airflow 16 approaches the end of the impeller disk 44. Referring now to FIG. 3, an exploded cross sectional view through the impeller 30 shows a series of impeller fuel channels 70 and a series of injection ports 72. The injection ports 72 inject the fuel 62 into a series of alternating passages 80 to form alternating streams of a rich fuel/air mixture 82 suitable for combustion and a dilution stream 84. The dilution streams 84 operate to reduce the temperature of the combustion products and to cool the turbine blades 46 of the rotating turbine system 20.

[0054] FIG. 4 shows a section through the tip of the impeller 30, which illustrates a combustor manifold 110, a series of combustor inlet ports 112 and a combustion chamber 114. The fuel/air mixture 82 is injected into the combustion chamber 114 where it is ignited and burned. The dilution stream 84 is injected along a series of pressure and suction surfaces 222, 224 of the turbine blades 46 where it isolates the blades from the high temperature combustion gases. A plurality of hot outflow gases 128 then flow radially inward through the turbine flowpath 60.

[0055] The torque required to compress the inlet airflow 16 is extracted from the rotating combustor gas turbine 10 by transmitting torque from the turbine blades 46 through the impeller disk 44 to the compressor blades 42. Because only a portion of the mechanical power generated by the rotating combustor gas turbine 10 is required to drive the rotating compressor blades 42, the excess power can be extracted by accelerating the hot outflow gases 128 to a high tangential velocity opposite to a direction of rotation R of the compressor/turbine impeller 30.

[0056] The outflow gases 128 are turned as they pass through a turbine nozzle 56[not shown]. This is accomplished by curving the trailing-edge 132 of the turbine blades 46 opposite the direction of rotation of the compressor/turbine impeller 30. As the outflow gases 128 pass the trailing-edges 132 they are accelerated by passing through a reduced throat area 134. This gives the outflow gases a high absolute tangential velocity leaving the compressor/turbine impeller 30. The high velocity outflow gases 128 are then diffused by the row of static radial exit guide vanes 26. The outflow gases 128 then exit the rotating combustor gas turbine 10 through the exhaust manifold 28 along an exit path 154.

[0057] The present invention is enabled by the ability to design a rotating combustion system 18 that can be used to burn the fuel/air mixture 82 in a very small volume over a short distance. Compression takes place from the inlet port 12 of the rotating combustor gas turbine 10 to a compressor exit 160 of the one-piece compressor/turbine impeller 30. Near the compressor exit 160, fuel 62 is injected and mixed with the inlet airflow 16 thereby creating the fuel/air mixture 82. The fuel/air mixture 82 then enters the rotating combustion system 18 where it is burned creating the hot outflow gases 128. The outflow gases 128 exit through the rotating turbine system 20, which serves the same purpose as the turbine stage in a conventional gas turbine.

[0058] From the perspective of the gas flowing through the rotating combustor gas turbine 10, the walls of the flow-paths appear to fixed relative to one another. This differs from a conventional gas turbine where the walls of the flow-path move relative to one another. This novel feature offers several distinct advantages over conventional gas turbine designs. First, this design reduces the number and the complexity of the components required to construct a gas turbine engine. Second, as all of the gas turbine components remain fixed relative to one another, the rotating combustion system 18 and the compressor/turbine impeller 30 can be

designed in such a manner as to reduce the heat load to the walls of the gas flow-paths and thereby reduce the wall temperatures for a given combustor outlet temperature, which is important because higher turbine inlet temperatures enable higher overall cycle efficiency and work output.

[0059] In the present embodiment, the fuel 62 can be introduced into the rotating combustion system 18 in such a manner as to generate large temperature gradients within the combustor exit/turbine inlet flow path 152. The temperature gradients within the combustor exit/turbine inlet flow path 152 are used to maintain the turbine blades 46 below the mean temperature of the gas flowing through the turbine system 20.

[0060] Prior art gas turbine engines employ turbine nozzle guide vanes to pattern the gas flow fields so as to lower the temperature of the gas in contact with the walls of the nozzles. However, in a conventional gas turbine design, the turbine blades move relative to the combustor and as a result the turbine blades are exposed to a circumferentially averaged mean temperature comprising intermittent high and low temperature gases. Therefore, because the turbine blades are exposed to the high mean temperature of the flow, expensive high temperature alloys are required in these systems while less expensive materials may be employed by the present invention to operate at similar temperatures. In addition, prior art gas turbine engines employ sophisticated internal and external cooling schemes to operate at temperatures above the maximum allowed by the blade materials. The ability of the present rotating combustor gas turbine design 10 to operate at mean temperatures above the maximum allowed by the material properties, without complex cooling schemes, represents a significant advance over conventional prior art gas turbine designs.

[0061] Another advantage of the present rotating combustor gas turbine design 10 is that the individual components (compressor, combustor and turbine) will produce lower losses than the components in a conventional design. The present invention includes the rotating compressor system 14, which is a common feature in small gas turbine designs. In a conventional radial compressor, as the flow leaves the impeller, the high speed of the flow is reduced using a diffuser. For high-pressure compressors this typically requires diffusers where the inlet flow is supersonic. Such diffusers are difficult to design and typically have low efficiencies.

[0062] In the present embodiment, the compressor efficiency is significantly improved by eliminating the need to diffuse this high-speed flow. Eliminating the need for a nozzle guide vane also reduces the losses in the turbine compared to a conventional turbine design. Also, as most of the components of the rotating combustor gas turbine 10 remain fixed relative to one another, few clearance gaps are required to allow the relative motion of the fixed and rotating parts. The compressor and turbine component efficiencies are improved by shrouding the flow path.

[0063] Conventional gas turbine combustors are designed to allow rapid, uniform and stable ignition of a fuel/air mixture; then to allow the necessary residence time for a series of chemical reactions to complete; and finally to allow the hot combustion products to be quenched using a bypass dilution air stream. Quenching is controlled to generate a desired combustor exit temperature profile that can be used to reduce the heat load on the turbine nozzle guide vanes. However, this must he carefully balanced against an increased mean midspan temperature that the turbine blades are exposed to. For the rotating combustor gas turbine 10 some of these functions are split between the rotating combustion system 18 and the rotating turbine system 20.

[0064] For the rotating combustor gas turbine 10, the primary function of the rotating combustion system 18 is to allow rapid, uniform and stable ignition of the fuel/air mixture 82. The residence time required for complete combustion is incorporated into the rotating turbine system 20, also a pattern factor is incorporated into the turbine flow path 60. Conventional gas turbine combustors generally use either bluff body or aerodynamic recirculation to create a region of slow moving, reverse flow where a small portion of the fuel/air mixture has enough time to react. In these systems, the hot gases in the recirculation flow are used to ignite an incoming fuel/air mixture, thereby ensuring stable combustion. These systems may also employ multiple small recirculation zones, which can be used to ensure rapid, uniform combustion of the incoming.

[0065] In the present embodiment, the rotating combustion system 18 is designed with centrifugal acceleration in mind, which together with the density gradients in the rotating combustion system 18 will have a significant impact on the combustor and turbine flow path 60. The cool inlet flows (fuel/air mixture streams 82 and the dilution streams 84) flow radially inward at a relatively high velocity. The incoming flows 82, 84 are cooler and denser

than the combustion flow in the combustion chamber 114. As the cool inlet flows 82, 84 are injected into the hot combustion chamber 114 they are rapidly decelerated by the centrifugal and Corriolis acceleration and forced to recirculate, thereby forming a recirculation zone 190.

[0066] This deceleration efficiently diffuses the inlet flows 82, 84 to a low combustor velocity thereby replacing the function of the diffuser in a conventional gas turbine. This allows for a high velocity flow upstream of the combustion system 18, which prevents flashback, or ignition of the fuel/air mixture 82 upstream of the combustion system 18. The recirculation zone 190, located downstream of the combustor manifold 110, recirculates the hot combustion products, which act as an ignition source for the incoming flow 82. These features combine to enable a very compact diffuser/combustor configuration that can be designed into rotating passages of a gas turbine.

[0067] The radial inflow turbine 20 is designed to perform some of the functions associated with the combustor in a conventional gas turbine. The turbine is designed to have a large flow area 210, which results in a very low radial flow velocity within the turbine passages. This low radial velocity increases the time duration that fluid particles spend in the turbine passage, when compared to turbine stages in conventional gas turbines. This increased turbine residence time ensures that there is enough time to complete the chemical reactions associated with combustion. In conventional gas turbines the combustion residence time is built into the combustor.

[0068] In certain embodiments of the present invention, as with diesel engines, the rotating combustor gas turbine 10 may require an alternate ignition mechanism to initiate combustion when the inlet temperature is too low for the compressor temperature rise to bring the combustor inlet flow 82, 84 temperature above the auto-ignition temperature of the fuel/air mixture 82. For example, the rotating combustor gas turbine 10 may use a series of resistive heating elements (not shown) mounted to the downstream side of the combustor manifold 110 and adjacent to the recirculation zones 190 to ignite the fuel/air mixture 82 under these conditions. The resistive heaters may be powered by an induction loop (not shown) positioned at the hub of the rotor and a stationary set of electromagnets (not shown) adjacent to the hub.

[0069] Other alternative ignition mechanisms include resistive heaters or spark ignition systems (not shown), which can be used to ignite the fuel/air mixture 82 in the rotating combustor system 18 for conditions under which the compression of the mixture would not lead to auto-ignition. The electrical current required to power the resistive heaters or the spark igniters can be delivered from the stationary housing to the rotating combustor using an induction loop, via slip rings, via a spark gap or by other similar means. Still other embodiments of the present invention can employ a catalyst such as platinum, which can be coated to a treated surface in the combustion chamber in order to reduce the temperature required to cause auto-ignition of the fuel/air mixture. Alternatively a pilot flame can be used to ignite the fuel/air mixture.

[0070] In addition to employing an alternative ignition system, the incoming air can be pre-heated to increase the rotating combustor gas turbine 10 inlet temperature so that the temperature rise in the compressor will enable auto-ignition of the fuel/air mixture. The incoming air can be pre-heated by combusting a small amount of fuel in a combustion chamber (not shown) in the impeller inlet flow path. Alternatively the incoming air can be preheated using an electrical heater. The inlet air pre-heater can be shut off once the combustor is lit.

[0071] In the present embodiment, the temperature of the turbine blades 46 is controlled by the dilution stream 84, which creates a sheath of cool air along the span of each blade 46, on both the pressure and suction surfaces 222, 224. This sheath isolates the blades 46 from the hot outflow combustion gases. In operation, there are large density gradients in the flow because of the temperature difference between the series of main flow and dilution streams. As the many streams flow radially inward, the centrifugal acceleration resulting from the rotation R of the compressor/turbine impeller 30 will decelerate the higher density low temperature flow, forcing the dilution streams to thicken and take up a greater proportion of the flow area 210.

[0072] In this embodiment, the core flow is forced to accelerate as its area is reduced, causing large gradients in velocity across the pitch of the passage. The gradients in velocity form a shear layer, which enhances the mixing of the streams. Because the radial inflow passage length is short relative to its width (and also relative to the thickness of the dilution streams), the mixing will not be completed when the flow exits through the turbine nozzle.

Therefore, temperature gradients from the pressure and suction surfaces 222, 224 to the center of the passage will isolate the turbine blades 46 from the high core temperatures.

[0073] The ability to maintain large temperature gradients (that are steady with respect to the walls of the flow passage) and to thereby isolate the turbine blades 46 from the high core temperatures by employing design features that are inexpensive to manufacture, gives the rotating combustor gas turbine 10 a significant advantage over conventional gas turbine designs. By comparison, it is extremely difficult and expensive to achieve such temperature gradients in the rotor flow path of a conventional high performance gas turbine rotor. The cooling (or dilution) air stream must be pumped through cavities in the rotor blades to small film-cooling holes where the coolant is injected into the mainstream. The blade cavities and film cooling holes are not only expensive to fabricate for conventional gas turbines, but are also impractical to incorporate in small gas turbines, as the manufacturing costs will actually increase at smaller scales.

[0074] An alternate embodiment of the invention is shown in FIG. 7. This embodiment of a rotating combustor gas turbine 250 uses an alternate turbine configuration, wherein other aspects of the invention are similar to those described in previous configurations. In this configuration a series of turbine blades 260 extend from a radius closer to an axis of rotation A-A, in contrast to previous embodiments wherein the turbine blades 46 extend from the combustor manifold 110 to the turbine nozzle 56, which leaves an annular combustion chamber 270 between the combustion manifold 110 and the turbine blades 260.

[0075] In this embodiment a series of combustion gases (not shown) flow radially inward and swirl tangentially in the direction of rotation R of the rotating compressor system 14. This configuration reduces the turbine surface area that must be cooled. To extract power from the flow the flow must still be turned tangentially, opposite to the direction of rotation R, and accelerated through a turbine nozzle throat 272. The core flow and the dilution streams mix before reaching the turbine blades 260. The turbine blades 260 may be internally cooled using compressor airflow, which is routed through a series of serpentine flow passages (not shown) in the turbine blades 260. Leading edge impingement cooling may also be used to control the temperature of the blades 260. Trailing edge ejection of the coolant flow can also be used to cool the turbine blades 260. These internal cooling features are common in internally cooled conventional turbine stages. The present rotating combustor gas turbine

configuration 250 benefits from a compact combustion configuration while relying on more conventional techniques to cool the hot turbine components.

[0076] Referring now to FIGS. 8-10, an alternate embodiment of the invention uses a radial outflow combustion and turbine configuration 300. Many features of this embodiment differ significantly from previously described configurations. For example the compressor flow path is not used as a heat sink for cooling the hot turbine components, therefore, its configuration is somewhat more simplified than previous embodiments.

[0077] In the present embodiment, incoming air flows through an inlet port 302, after which it flows through a series of pre-swirl vanes 304. The pre-swirl vanes 304 spin the flow in the direction of rotation of an impeller 306. The flow then enters the radial impeller 306, where it flows radially outward. The walls of the flow path are comprised of a series of impeller blades 310, an impeller disk 312 and the impeller shroud 314.

[0078] A fuel 320 flows through a fuel inlet port 322 and flows onto the impeller 306 through a fuel nozzle 326. In the impeller 306 the airflow is separated into a core flow, which travels through a core flow path 330 and a cooling flow, which travels through a cooling flow path 332. The fuel 320 then flows radially outward through a series of fuel channels (not shown) on the impeller 306 and is injected into the core flow path 330 where it mixes with the core flow. The core flow, which comprises a fuel/air mixture suitable for combustion, flows into a combustion chamber 340 through a plurality of small combustor inlet ports 342 in a combustor inlet manifold 344.

[0079] The combustion chamber walls 346 are cooled by routing the cooling airflow stream through a plurality of coolant inlet ports 348 in the combustor inlet manifold 344. The coolant stream flows through a series of serpentine cooling passages, which increase the cooling effectiveness of the flow. The serpentine flow passages are made from combustor manifold walls 354, which protrude from the combustor manifold 344. The walls of the serpentine coolant flow path are comprised of the combustor manifold 344, the combustor manifold walls 354 and the turbine nozzle 348. The serpentine cooling flow also cools a series of nozzle walls 348.

[0080] At a combustor manifold exit 362, the core flow and the coolant flow are combined together close to the nozzle throat 360. The coolant flow isolates the turbine nozzle walls

348 from the high temperature gas in the core flow as it exits from the combustor manifold 346. After leaving the nozzle 348 the flow then enters a row of static radial exit guide vanes 364, which diffuse the high velocity flow leaving the nozzle 348. The power generated in the rotating combustor gas turbine 300 is extracted through a shaft 370.

[0081] In an alternate embodiment, a radial outflow combustor uses an axial outflow turbine nozzle configuration. While most features of this embodiment are common to the radial outflow combustor configuration described above, this alternate configuration uses a turbine nozzle configuration that turns the flow to an axial direction before leaving the rotor. In the radial outflow configurations described above the flow leaves the nozzle with a high tangential velocity and a relatively low radial velocity. This alternate embodiment turns the flow from radial to axial before leaving the nozzle so that the flow leaving the nozzle has a high tangential velocity and a relatively low axial velocity. This turbine configuration enables a simpler, more compact multi-stage design. As an alternative embodiment, an axial outflow nozzle configuration can also be used with a radial inflow combustor configuration.

[0082] In certain gas turbine engines that employ premixed fuel air mixtures, the flow velocity must be kept high enough to prevent flashback under all operating conditions. Also, the premixed fuel air stream cannot be used as a heat sink to cool the combustor and turbine walls, as the hot walls could cause the mixture to ignite before entering the combustion chamber. Therefore, certain embodiments of the invention, which are more suitable for use with, either premixed fuel/air mixtures, or for liquid fuels that require additional time to allow the fuel to vaporize and mix with the core air are described below.

[0083] Referring now to FIGS. 11-14 alternative embodiments of the present invention are disclosed. These additional embodiments utilize a main impeller 380 and a secondary impeller 382, to isolate the premixed fuel air mixtures from the heated main impeller walls 390, 392. This secondary impeller 380 utilizes a series of radial compressor blades 400, to compress the flow. Referring now to FIG. 13, a section through the main impeller 380, the secondary impeller 382 and an impeller manifold 396 is shown. The core flow, which contains either a premixed fuel/air mixture, or an air stream with an atomized fuel in the process of vaporizing, flow through a series of secondary impeller channels 410, while a cooling air flows through the a series of primary impeller passages 412.

[0084] FIG. 14 shows an embodiment of the present invention suitable for use with premixed fuel/air mixtures. A core flow, which contains a fuel/air mixture suitable for combustion flows through an outer channel 420 and a cooling air flows trough an inner channel 422. FIG. 14 further shows the flow path between a tip of the impeller 424[not shown?] and a combustor manifold 426[not shown?]. The combustor manifold 426 transitions the flow from an axial segmented flow in the impeller to a circumferentially segmented flow required in the combustor.

[0085] Referring now to FIGS. 15 and 16, in certain embodiment of the present invention, the outer portion of the turbine shroud 52 is cooled by utilizing disk windage. The hot gases in the rotating turbine system 20 have a low velocity relative to the turbine shroud 52. A downstream side 58 of the turbine shroud 52 is exposed to an engine cavity 66, which has a relatively low temperature. However, relative to the turbine shroud 52 the air in engine cavity has a high tangential velocity. This cool high velocity flow cools the turbine shroud 52 and keeps its temperature within allowable material limits.

[0086] To ensure that the air in the engine cavity 66 is relatively cool the secondary airflow paths must be controlled. In order to allow relative motion between the moving components such as the turbine blades 46 and the stationary components such as the static radial exit guide vanes 26, a gap 38 is required. A seal 48 controls the flow of gas out of, or into the main gas path. In conventional gas turbine engines the seals are very precise, expensive parts. In several embodiment of the present invention, the pressure ratio, exit temperature, and turbine nozzle exit Mach number can be designed in such a manner that the static pressure in the gap 38 is somewhat below the engine cavity pressure. This will cause a small inflow of gas from the engine cavity into the primary gas path, thereby purging the engine cavity 66 and keeping its temperature relatively cool.

[0087] In certain embodiments, the gases in the turbine nozzle 56 are accelerated to a high velocity relative to the turbine shroud 52, which results in high levels of heat transfer between the hot gas and shroud 52. In these embodiments, a secondary flow path of cool air from the engine cavity 66 can be used to provide additional cooling. To create the secondary flow path, a series of secondary impeller blades 74, which protrude from the turbine shroud 52, pump cool air from the engine cavity 66 and enhance heat transfer to the turbine shroud 52.

A secondary shroud 76 covers these impeller blades 74. The secondary cooling flow may then be discharged back into the engine cavity 66.

[0088] In yet another embodiment of this invention, which can be used in conjunction with several of the afore-mentioned embodiments, is a cooling system (not shown) designed to cool the walls of the rotating combustor and nozzle. The cooling system allows the wall temperatures of the combustor and nozzle to operate below the temperature of the adjacent gas flow path. The cooling system comprises a closed loop through which a liquid, preferably a liquid alkali metal, flows. Heat is transferred from the hot gas to the walls of combustor and nozzle flow path and then to the circulating liquid. The liquid then flows radially inward to a rotating heat exchanger located upstream of the combustor. Heat is then transferred from the liquid to the walls of the heat exchanger and to the incoming flow. The pressure differential required to pump the liquid is provided by the combination of the density differential that results from the temperature change in the liquid and the centripetal acceleration caused by the rotation of the system. The lower operating temperature enables the rotating components to be constructed from a broad range of inexpensive materials.

[0089] An alternative embodiment of the present invention includes a cooling system, which utilizes a cooling fluid that undergoes a liquid-vapor phase change, whereby the fluid in contact with the hot walls of the combustor and nozzle vaporizes. The vapor then flows to the heat exchanger (not shown) upstream of the combustor where it condenses back into a liquid and flows radially outward back to the combustor and nozzle.

[0090] In addition to operating as a stand-alone engine system, several embodiments of the rotating combustor gas turbine 10 can be used as the core of a multi-stage gas turbine design as illustrated in FIG. 17. Single or multiple additional compressor stages (radial or axial) can added upstream of the rotating combustor gas turbine 10 to increase the overall engine pressure ratio. Single, or multiple additional turbine stages (axial or radial) can be added downstream of the rotating combustor gas turbine 10 to expand the high-pressure high temperature gases. The additional compressor and turbine stages and the rotating combustor gas turbine 10 can rotate on a common shaft 440 or on separate spools.

[0091] In the present embodiment, an alternate rotating combustor gas turbine configuration utilizes a conventional radial compressor 460 upstream of a rotating combustor gas turbine

450 to increase the overall cycle pressure ratio. Compression of the inlet flow increases its temperature; high compression ratios will increase the airflow temperature so that the fuel/air mixture will auto-ignite, as is common in diesel engines. Ignition delay prevents the fuel/air mixture from igniting upstream of the combustor. The recirculation zones in the combustor give the fuel/air mixture the required residence time to allow ignition to occur. An uncooled axial flow turbine 470 is shown downstream of the rotating combustor gas turbine 10.

[0092] Alternative embodiments of the present invention include a counter-rotating turbine. In certain of these embodiments, a counter-rotating turbine stage (not shown), such as a conventional turbine that is rotating in the opposite direction to the gas turbine engine 10, may be used downstream of the turbine nozzle 56 in place of the set of stationary exit guide vanes 26.

[0093] In addition, the gas turbine engine 10 may employ a rotating nozzle (not shown) that utilizes a convergent-divergent geometry. This would enable supersonic Mach numbers downstream of the turbine nozzle 56, which may be more suitable to a counter-rotating turbine configuration.

[0094] In an alternative embodiment, a gas turbine engine includes a separate compressor, a separate combustor and a separate turbine, wherein these components are operatively assembled together to make a rotating combustor gas turbine engine. This gas turbine engine allows individual engine components to be manufactured, removed and or replaced separately as well as allows additional or alternative cooling systems to be employed. A nozzle impingement cooling system, which makes use of leakage flow in the gas turbine engine, may provide enhanced cooling to the areas of the nozzle that have he highest heat load.

[0095] Referring now to FIG. 18 , an alternative structural configuration of a rotating combustion gas turbine 600 is shown. This configuration is designed to isolate a load bearing impeller disk 680, and an aft disk 682 from a series of high temperature combustion gases 572. This has several benefits, namely; this isolates a compressor flow path 534, from the high temperatures thus preventing pre-ignition of the fuel/air mixture. Maintaining the load bearing structures 680, 682 at a relatively low temperature will enable the use of materials such as titanium that may allow higher impeller tip speeds.

[0096] The stresses in the high temperature combustor 610 and turbine structures 552, 553 can be reduced by transferring some of the centrifugal load from these parts to impeller and aft disks 680, 682. This may in turn enable the use of high temperature ceramic materials that have relatively poor tensile properties. A pair of aft and forward end-walls 552, 553 of the turbine flow path are maintained at a low temperature by allowing a controlled amount of secondary flow 660 to flow between a stationary aft shroud 670 and a forward shroud 672. The secondary flow 660 flows from the engine cavity 690 then flows into a turbine exit flow path 560. Alternatively hoops of high strength composite materials (not shown) can be used to hold some of the centrifugal load of the high temperature combustor 610 and turbine structures 552, 553.

[0097] Although the preferred embodiments of the present invention have been described herein, the above descriptions are merely illustrative. Further modifications of the invention herein disclosed will occur to those skilled in the respective arts and all such modifications are deemed to be within the scope of the invention as defined by the appended claims.